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Probabilistic Analysis of an HSCT Modeled with an Equivalent Laminated Plate Wing

Dr. Dimitri N. Mavris
Assistant Professor & Manager, ASDL

William T. Hayden
Graduate Research Assistant, ASDL

Aerospace Systems Design Laboratory, School of Aerospace Engineering, Georgia Institute of Technology

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ABSTRACT

The High Speed Civil Transport (HSCT), a supersonic commercial transport currently under development, presents several challenges to traditional conceptual design. The current historical database used by many commercial transport design processes only include data for subsonic transports and therefore does not apply to innovative new configurations such as the HSCT. Therefore, physics-based, preliminary design tools must be used to model the characteristics of advanced aircraft in conceptual sizing routines. In addition, the evaluation of the aircraft design space often requires the analysis of many configurations in order to assess the impact of design constraints and determine the attainable range of system level metrics, a process which is very time consuming in both modeling and computer run time.

To address these challenges, the equivalent plate structural analysis code ELAPS is used to model the wing structure of the HSCT and the Fast Probability Integration (FPI) technique is used to probabilistically assess the design space. After the ELAPS model is generated in a parametric manner, the structure is optimized to yield a weight for each component of the wing. A Response Surface Methodology approach is then implemented using Design of Experiments tables and an analysis of variance to generate response surface equations in terms of the most influential design variables for these wing component weights, as well as for the fuel volume available. These expressions are substituted into the sizing and synthesis code FLOPS in order to conduct system level design trade studies. FLOPS is subsequently enhanced with equations created from physics-based tools for the various disciplines to create a preliminary design synthesis tool. Ranges for the system level design variables are then introduced through the FPI technique, a probabilistic process which generates cumulative distributions for system level metrics such as take-off gross weight. This technique requires only 20 to 30 executions for FLOPS to generate these distributions, hence greatly reducing the time required to conduct the analysis.

The results of this study indicate that the HSCT has only a 20% probability of achieving the system level design constraint of a 1,000,000 lb. take-off gross weight with the current level of technology and has no chance of achieving the desired goal of 750,000 lb. Through the use of new enabling technologies, however, these weight levels can be reduced to increase the probability of achieving technical feasibility and improve its economic viability. Future efforts will therefore focus on the evaluation of these technologies and their impact on system level performance and economics.

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INTRODUCTION

Over the next 10 years, trans-Pacific air travel will nearly double in volume[1]. However, current subsonic transports often require 14 hours on many of these routes such as Los Angeles to Sydney. Therefore, the need and market exists for a supersonic transport which can cut this time in half. In response to this need, NASA and several members of the U.S. aircraft manufacturing industry are engaged in the High Speed Research program (HSR), a study which will bring to maturity the technologies required to build an economically viable High Speed Civil Transport (HSCT) by midway through the next decade.

The HSCT is envisioned to travel at Mach 2.4, have a range of 5000 NM., and carry approximately 300 passengers[1]. However, it must also conform to existing performance standards and regulations in effect for the subsonic fleet. As a result, many design issues have surfaced during the development of this aircraft. In particular, the structural modeling and sizing of an HSCT configuration has required the development of new tools and processes, many of which are extremely complex and require months to complete one design cycle.

These new approaches to structural modeling have become necessary due to the unusual configuration of the HSCT as shown in Figure 1. Many conceptual level structural weight estimations for subsonic transports are based on historical data of similar configurations. This data becomes invalid for new concepts such as the HSCT, thus a new database for structural weight estimation is needed for this approach. However, since data only exist for a few supersonic transports, such as the Concorde and the Tu-144, structural weight estimates can no longer rely on historical databases. Therefore, a great deal of effort has been expended to developed a physics based model of the HSCT structure in order to derive representative weights for conceptual structural sizing.

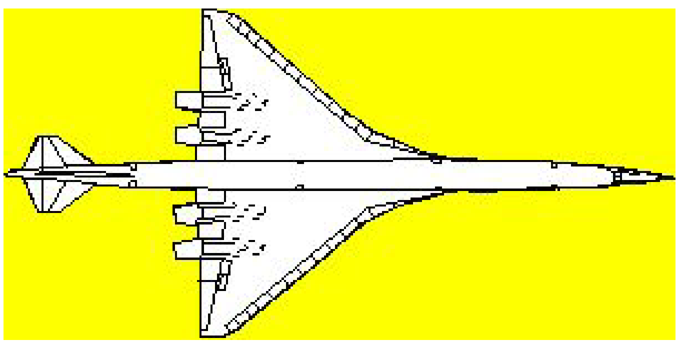

Figure 1: Typical HSCT Configuration[2]

Another challenging problem worthy of attention is the determination of means for the rapid assessment of the impact of constraints on the design space. In order to generate a representation of the design space, the various design variables must be varied systematically within their ranges, a combinatorial problem needing 100s - 1000s of permutations. Therefore, in order to calculate the resulting distributions for system level metrics such as structural weight, hundreds of configurations would have to be modeled. However, since physics based methods must be used to model the HSCT, much time is required to generate structural models for each configuration. As a result, an alternate method for generating these distributions must be found in order to evaluate the probability of satisfying system level design constraints .

Fortunately, new methods in structural modeling and thus corresponding probabilistic assessment have been developed to address these issues. Advances in equivalent plate structural analysis offer both reduced model generation time and reduced computational computer run time for wing structures. In the area of probabilistic analysis, new approaches from the field of structural reliability have allowed the generation of cumulative probability distributions from very few solution runs. This study will therefore focus on the application of these new techniques to the HSCT design challenge.

IMPLEMENTATION

The process of aircraft design is currently shifting its focus to a reduction in design cycle time in order to reduce costs and deliver the product to market in a timely manner. However, design concepts such as the HSCT are challenging because they lie outside the historic database and demand a thorough physics based analysis in order to accurately model the characteristics of the aircraft. The time required to conduct this analysis as well as a probabilistic analysis of the HSCT design space is significantly reduced, though, through the use of Equivalent Laminated Plate Solver (ELAPS) for structural modeling, FLight Optimization System (FLOPS) for vehicle sizing and synthesis, and the Fast Probability Integration (FPI) technique to probabilistically evaluate the design space.

ELAPS

ELAPS[3,4], developed by Dr. Gary Giles at NASA Langley Research Center, utilizes equivalent laminated plate theory to simplify wing models. This process essentially models the wing as a plate of equivalent thickness. Internal structure is introduced as an additional thickness smeared over the plate. The kinetic and potential energy of the structure and forces are calculated assuming that the wing deflections are composed of a combination of wing displacement functions. Then, using the Ritz solution, the deflections and stresses are computed as continuous functions. In addition, generalized masses and stiffnesses as well as natural frequencies and mode shapes are determined.

This technique offers many advantages over conventional finite element models of wing structure. First, the time to generate the model is significantly reduced due to the greatly simplified geometry of an equivalent plate wing. This technique also requires up to 50 times less computer run time than finite element solutions[5]. The continuous definitions of stresses and deflections offer the advantage of easy coupling with aerodynamics codes for aeroelastic studies as well. In addition, this technique comes within 5% of finite element solutions for deflection, stress, and mode shape calculations[5]. These advantages clearly warrant the investigation of these techniques for the modeling of HSCT wing planforms.

However, without the ability to optimize a structure for a specified load case, these techniques are of little use in design. During structural optimization, the thickness of each structural member is varied in order to minimize the weight of the structure while still maintaining the strength required to support the applied load. This process must be carried out on each configuration under review in order to truly assess the benefits of each design. Due to the fact that ELAPS does not currently incorporate a structural optimization feature, a first order approximate method is applied and incorporated into ELAPS to address these issues.

FLOPS AND RESPONSE SURFACES

In order to evaluate the HSCT system as a whole, a vehicle sizing and synthesis code is required. For this study, FLOPS[6] is used to size the vehicle for a given set of mission requirements, constraints, and design variable ranges at a given level of technology. For analysis of subsonic transports, FLOPS contains equations which are based on historical databases for aircraft of this type. However, as mentioned previously, an appropriate historical database does not exist for an HSCT configuration. Therefore, through the use of the Response Surface Methodology (RSM)[7,8] approach, new weight equations for the wing, based on ELAPS analyses of HSCT wing planforms, have been developed and incorporated into FLOPS.

RSM incorporates a Design of Experiments (DoE)[8,9] approach, in order to formulate a polynomial equation for a given response in terms of the appropriate design variables. A screening test utilizing a Pareto analysis is used to identify the variables which are the most significant contributors to response variability[10]. This procedure is illustrated in Figure 2.

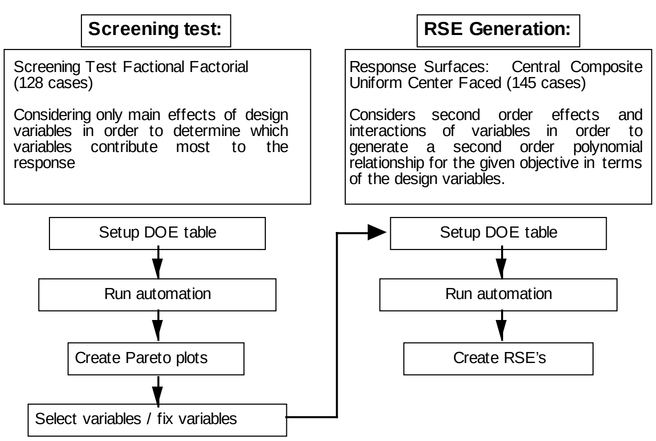

Figure 2: RSM Implementation Procedure

For this study, the polynomial in terms of these variables is second order of the following form:

$$R = b_o + \sum_{i=1}^{k} b_i x_i + \sum_{i=1}^{k} b_{ii} x_i^2 + \sum_{i=1}^{k-1} \sum_{j=i+1}^{k} b_{ij} x_i x_j \qquad (1)$$

where the b_i represents the regression coefficients for the linear terms, b_{ii} the quadratic coefficients, b_{ij} the cross-product coefficients (i.e. second order interactions), x_i and x_j the design variables, and $x_i x_j$ denotes interactions between two design variables. Once EQ (1) has been determine for the weight of the HSCT wing through the use of ELAPS, it can be used in FLOPS to replace the historically based wing weight equations. In this way, the characteristics of the HSCT are accurately modeled in the synthesis and sizing code without the use of time consuming finite element models. In addition, a similar approach is used to incorporate aerodynamic and acoustic information into FLOPS for HSCT specific configurations, but this discussion is beyond the scope of this paper.

FPI

Once FLOPS is converted to an HSCT specific synthesis and sizing code through the use of response surface equations, a probabilistic analysis of the design space is conducted. For this study, the FPI[11,12] technique, developed by the Southwest Research Institute, was employed to generate cumulative frequency distributions of system level responses such as take-off gross weight without having to analyze an excessive number of configurations. This technique evolved in the field of structural reliability, where the overlap of strength and stress of a material can lead to failure. To represent the effect of uncertainty in structural configuration and material properties, probability distributions were used to model these quantities. The result was a distribution for material strength and stress, with the overlapping portions of these distributions representing the probability of structural failure. However, in order to generate these distributions through a standard Monte Carlo[13] approach, hundreds or possibly even thousands of finite element models had to be generated and analyzed. This approach is obviously impractical due to excessive computer run times.

Therefore an alternative approach was developed. By linearizing the relationship between the design variables and the response, the design space can be normalized and the probability of achieving a certain value for the response can be determined. The analysis is then run again without the linear approximation to determine the true value of the response that corresponds to the calculated probability. In this manner, cumulative frequency distributions can be generated for a response with as few as 20 executions of the analysis code. Therefore, a probabilistic assessment of the design space can be performed without the costly computer run times of a Monte Carlo approach.

For this study, the FPI technique is applied to FLOPS at the system level. Ranges for design variables such as configuration geometry are introduced as uniform probability distributions and their effects on system level responses are determined through the generation of cumulative frequency distributions. Therefore, through the use of ELAPS, RSM, FLOPS, and FPI, this study presents the results of a design study of an HSCT configuration incorporating accurate structural modeling and a probabilistic assessment of the aircraft design space, as well as illustrating a dramatic reduction in computational time required to complete the analysis.

APPROACH

The process by which ELAPS structural modeling and FPI probabilistic analysis are applied to the HSCT design process is shown in Figure 3. First, an HSCT wing is modeled parametrically in ELAPS and structurally optimized for a critical load condition. This process is automated to reduce design cycle time. Through the use of a DoE formulation, response surface equations for the weight of the strake, inboard wing box, and outboard wing box are then determined in terms of design variables such as wing geometry. The RSE's are adjusted to account for non-modeled weight, non-structural weight, and various material concepts using information from previous HSCT design studies. These equations are then integrated into the FLOPS synthesis and sizing tool in place of the historically based regression equations. The FPI technique is then used to represent the design variables as uniform probability distributions in order to generate a probabilistic representation of the design space. A cumulative distributions for the vehicle take-off gross weight, a key system level metric is then calculated.

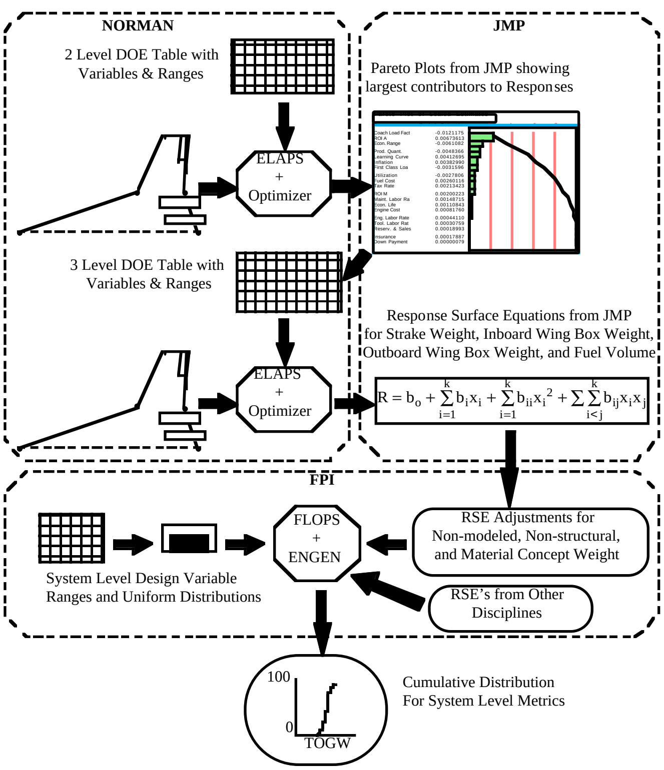

Figure 3: Procedure for the Implementation of ELAPS and FPI in the Design Process

Take-off gross weight (TOGW) has historically been the key metric in transport aircraft design, with emphasis always on minimization. Although any HSCT configuration developed today must first and foremost be economically viable and therefore dependent on economic metrics such as required average yield per revenue passenger mile ($/RPM), the TOGW of the aircraft will continue to be one of the primary driving forces in its structural design. Not only will a lower TOGW configuration require less fuel to complete a given mission, an operational ceiling exists on this metric due to the existing airport infrastructure. Current runways can generally handle aircraft up to 1,000,000 lb., depending on the configuration of the landing gear. Therefore, this weight can be thought of as a constraint on the design, with any configuration over 1,000,000 lb. technically infeasible. In addition, based on previous studies, the authors have come to the conclusion that aircraft weights in the 700,000 lb to 750,000 lb category are desirable for an economically viable solution. Therefore, with this constraint and target superimposed onto the cumulative frequency distribution for TOGW, the probability of achieving a technically feasible design or of achieving the target weight can be determined.

MODELING WING CONFIGURATIONS IN ELAPS

The wing of the HSCT is the only structural component to be modeled in ELAPS in this study. This planform must combine the characteristics of a delta wing for supersonic cruise while maintaining the low speed performance of a higher aspect ratio wing for take-off and landing. Therefore, the design is compromised aerodynamically due to these competing considerations. However, previous studies have shown that a double delta or arrow wing planform would be a good compromise between high speed and low speed performance.

In a typical design study, only one wing configuration is studied at a time. By modeling the wing parametrically, though, many configurations can be examined simultaneously. This parametric definition of the HSCT wing is shown in Figure 4 with the ranges for these variables listed in Table I.

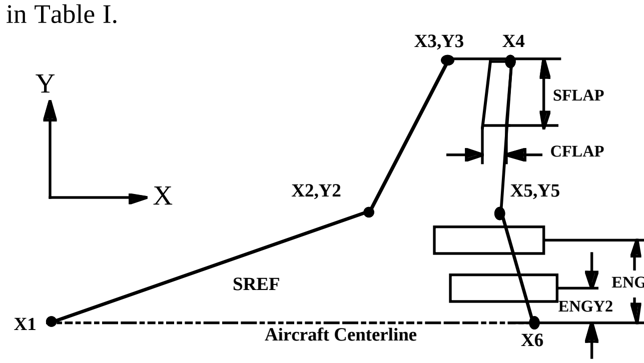

Figure 4: Parametric Model of HSCT Wing Planform

Table I: Design Variable Ranges for the Parametric Modeling of the HSCT Wing Planform

Variable	Min.	Mid Point	Max.
X2	1.54	1.615	1.69
Y2	0.44	0.51	0.58
X3	2.1	2.23	2.36
X4	2.4	2.49	2.58
X5	2.19	2.275	2.36
Y5	0.44	0.51	0.58
X6	2.19	2.345	2.5
S (ft^2)	7000	8300	9500
CFLAP	0.25	0.3	0.35
SFLAP	0.15	0.2	0.25
GW (lb)	700000	850000	1000000
ENGY1	0.24	0.27	0.3
ENGY2	0.49	0.52	0.55
ENGWT (lb)	10000	16000	20000

(X and Y values non-dimensionalized by semi-span)

By defining the wing in this manner, the response surface equations generated for the weight of the various wing components will be valid for any number of configurations. Figure 5 shows a sample of these configurations to illustrate the range of validity for these expressions.

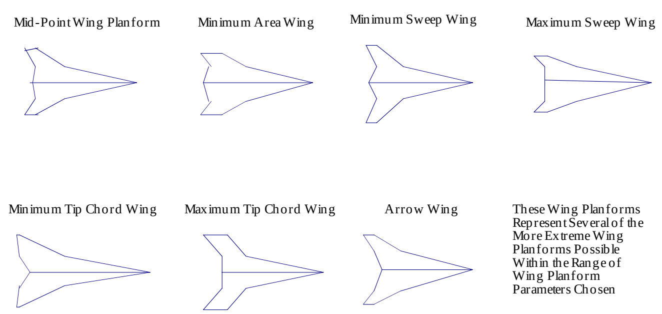

Figure 5: A Sample of Planforms Which Are Captured Through Parametric Modeling of the HSCT Wing

Assumptions

In order to allow complex wing configurations to be modeled in ELAPS several assumptions were made in the analysis. These included the wing structural material, boundary conditions, optimization criteria, and weight adjustment for non-modeled factors. The assumptions used in the modeling of these parametric planforms in ELAPS are listed as follows:

- Wing modeled in ELAPS as plate of equivalent thickness
- Titanium used for ELAPS analysis (Ti-6Al-4V, ρ=0.160 lb./in^3, E=16x10^6 psi, G=6.2x10^6 psi, γ=0.36)
- Minimum gauge of internal structure 0.12 in.
- Wing cantilevered at root
- Wing structure sized by bending stresses only due to analysis limitations
- Optimization based on beam bending stress
- ELAPS output weight scaled to account for non structural weight, materials and structural concepts from information provided by 1975 Lockheed SST study[14]

Loads

In addition to structural modeling considerations, the loads which are to structurally size the model must be considered as well. For a typical double delta configuration, a 2.5 g pull-up maneuver at Mach 0.9 is the critical condition for the upper skin panels of the inboard wing box. The lower skin panels are sized similarly by a pull-down maneuver. Flutter constraints typically size the structure of the outboard wing box, rather than strength considerations. Finally, the strake region of the wing is generally minimum gage due to a lack of high bending stresses and low impact on flutter stiffness. Due to the constraints of the analysis approach presented here, though, only the 2.5 g pull-up maneuver will be used for structural sizing. These loads are generated in VORLAX[15], a vortex lattice potential flow code, and are transferred to the ELAPS model. Furthermore, the following assumptions are also made concerning the loads modeling for the HSCT wing planform:

- Potential flow aerodynamics used to calculate loads
- Loadcase: 2.5 g pull-up maneuver at M=0.9
- Loads parametrically varied with configuration
- Wing sized with full fuel loads
- Engines modeled as point masses on ELAPS model
- Landing gear modeled as point mass sized by gross takeoff weight

It is worth noting that the last three bullets in this list not only contribute to the weight of the model but also are related to a concurrent flutter analysis which was also conducted on these planforms. This is especially true of the decision to generate models with full fuel loads, since empty fuel conditions generally are the critical cases for strength sizing. However, due to the limited scope of the structural sizing and the need to generate models which can also be analyzed for critical flutter cases, the wing was modeled for the full fuel condition. The results of the flutter analysis, however, are beyond the scope of this study.

Automation

In order to implement the DoE approach for generating response surface equations, approximately 150 wing configurations have to be modeled and structurally optimized in ELAPS. Therefore, this process must be automated in order to take advantage of the time savings ELAPS offers, even when analyzing numerous configurations. The goals for this automation process must be to generate model planform geometry, thickness distribution, internal volume, and loads to feed to the ELAPS input files. In addition, the process must also conduct the structural optimization and gather the results in a format for the analysis of variance which will generate the response surfaces. A schematic of this process is shown in Figure 6.

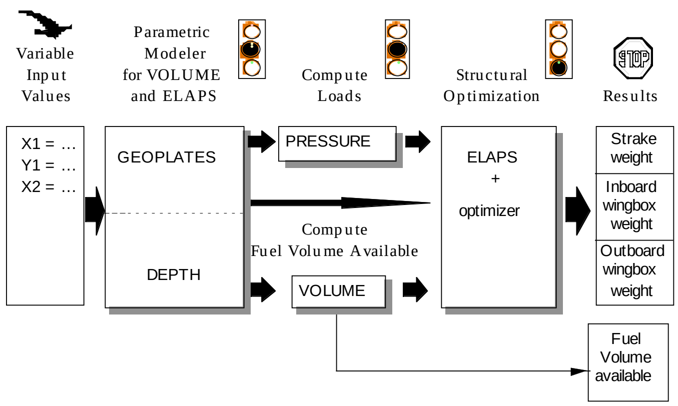

Figure 6: Automation Process for ELAPS Wing Model Generation

Each stage of this automation procedure is accomplished by separate FORTRAN codes. In the first stage, the model geometry is translated from design variables to the coordinates of the 14 plates which comprise the ELAPS model. In addition, the engine and landing gear locations are determined. This information is then passed to the second and third stages. The thickness distribution of the wing is computed in the second stage using non-dimensionalized definitions of the airfoils. For this study, the airfoils inboard of the leading edge kink are subsonic which the airfoils outboard of this kink are supersonic. The third stage computes the volume available in the wing for fuel by discretizing the wing into small boxes, computing the volume of these boxes using the algorithm of stage two in conjunction with the box dimensions, and sums these boxes over the planform of the wing. This stage is primarily used to generate response surfaces for fuel volume available in the wing for later integration in FLOPS. The loads which are computed by VORLAX are applied to the model in stage four, adjusting for vehicle weight to achieve a 2.5 g maneuver and planform geometry. The execution of these stages and the passage of information between them is accomplished through the use of NORMAN[16], an automation tool often employed for computational DoE work. NORMAN is also used to control ELAPS execution and parsing of results. The final results of these stages of the automation procedure, however, are the ELAPS input files for the various wing configurations in the DoE table which are now ready for analysis and structural optimization in ELAPS.

STRUCTURAL OPTIMIZATION

As discussed previously, two approaches to structural optimization are attempted in this study. In the first, the ADS[17] optimizer, developed by Dr. Vanderplaats at NASA Ames, is linked with ELAPS. This optimizer is set to employ sequential linear programming, the method of feasible directions, and the Golden section method for the 1-D search in this analysis. However, due to difficulties encountered with this approach, including long computer run times due to numerous executions of ELAPS and inconsistent sensitivities of resulting weights to changes in design variables, it was abandoned in favor of a first order approximation.

The theory behind this first order approximation is found in 2-D beam theory. The individual plates which compose the skin of the wing are modeled as beams. In this study, the structure is only optimized for bending stress, allowing this assumption to be made. Therefore, the simplified equation for bending stress in a beam is shown in EQ (2):

$$\sigma = My / I \tag{2}$$

where σ is the bending stress, M is the bending moment applied to the beam, y is the distance from the neutral axis of the beam to the point at which the stress is being calculated, and I is the area moment of inertia of the beam. Treating the skins as beams, y and I become:

$$y = t / 2 \tag{3}$$
$$I = bt^3 / 12 \tag{4}$$

where t is the thickness of the skin and b is the base length. Now, if the thickness of the plate was optimum, the stress in the plate would be the yield stress plus a safety factor (which will be referred to as σ_{yield}), resulting in EQ (5):

$$\sigma_{yield} = M(t / 2) / (bt_{opt}^3 / 12) \tag{5}$$

For any other thickness, the equation has the form shown in EQ (6):

$$\sigma = M(t / 2) / (bt^3 / 12) \tag{6}$$

Dividing the first of these two equations into the second and rearranging the terms yields EQ (7):

$$t_{opt} = t(\sigma / \sigma_{yield})^{1/2} \tag{7}$$

Notice that this equation assumes that the load M is constant for both equations and that the base length b does not change as well. This equation is only a first order approximation to a very complex problem. However, EQ (7) is found to give excellent results when used in an iterative fashion.

This process was incorporated into the ELAPS source code for implementation. Results of testing showed that convergence is generally achieved with between 10 and 20 executions of ELAPS, as compared to the nearly 1500 executions often required by the ADS optimizer. However, this large number of executions is usually the result of gradient calculations for a large number of design variables. In addition, the sensitivities of the resulting wing weights to changes in the design variables showed consistent trends for the variable ranges used in this study.

For this analysis, the thickness of the wing skins are the variables being optimized with the ultimate stress of the material acting as the constraint and the minimization of the weight of the structure the objective. The thickness of the internal members is set to a minimum gage due to the repeated tendency of these thicknesses to optimize to extremely small values because of the neglect of shearing loads. However, these members are still included in the model weight. A schematic of the internal layout of the wing structure is shown in Figure 7.

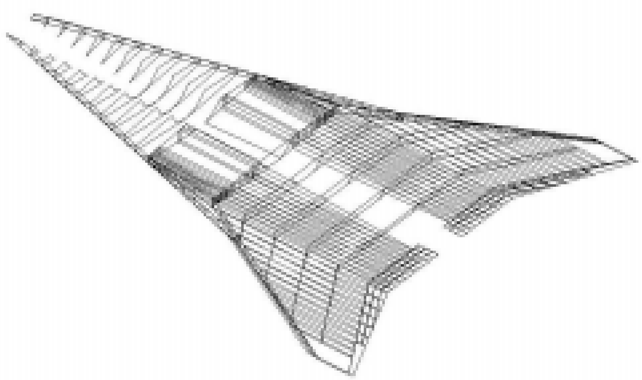

Figure 7: Internal Structure of HSCT ELAPS Wing Models

With the integration of this optimization scheme with ELAPS, the automation scheme for determining the optimized wing weight from any combination of design variables is complete.

SCREENING TEST

At this point, a screening test is performed to determine the design variables whose variance contributes most to the wing weight. This is accomplished by executing a two level fractional factorial DoE with 128 cases using the JMP[18] program. NORMAN is used in conjunction with the automation and optimization scheme discussed previously to execute each of these 128 cases and gather the resulting weights for the strake, inboard wing box, and outboard wing box (as well as the fuel volume available). With this information, an ANalysis Of VAriance (ANOVA) is performed to generate a linear relationship between the design variables and the responses. In addition, Pareto plots are generated which rank the variables in order of impact of their variance on the chosen response. A sample Pareto plot is shown in Figure 8.

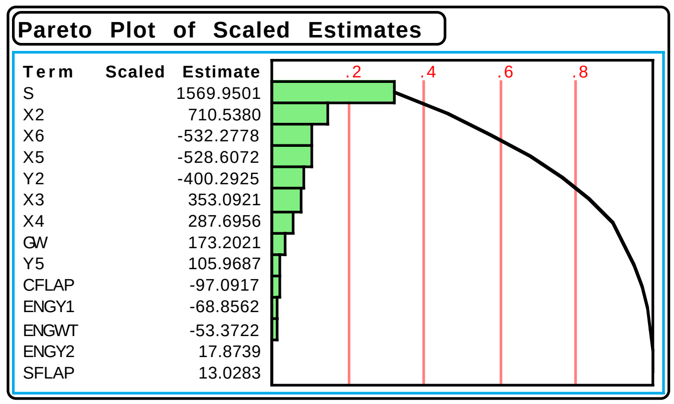

Figure 8: Pareto Plot for the Weight of the Strake

With this DoE, only the main effects of the variables are considered in order to determine those that are most statistically significant. The objective is to identify the variables that contribute at least 80% of the response. However, as shown in Figure 7, the first eight variables contribute over 90% of the response. Since the DoE for an eight variables response surface equation calls for 145 cases, with the number of cases increasing quickly with each additional variable, only the top eight variables for each response are chosen for the response surface DoE. These variables are shown in Figure 9.

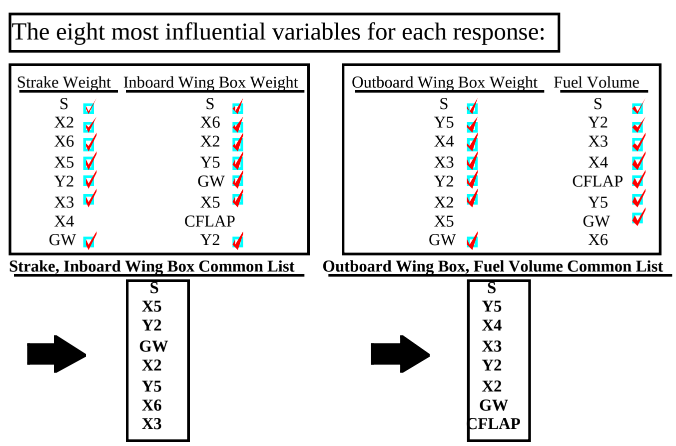

Figure 9: Screening Test Results for Wing Component Weights and Fuel Volume Available

Inspection of Figure 9 indicates that the engine locations and engine weight no longer appear on these lists of this design variables. This omission is due to the modeling of the engines as concentrated masses for flutter analysis purposes. However, this representation of the engines does not contribute to the internal stresses of the structure. Therefore, these variables showed no effect on the optimized weight of the structures and are omitted for the remainder of the analysis. The variables that do contribute most to the response are the wing area and kink location in the spanwise direction. Wing area is dominant not only due to the obvious direct correlation between this variable and wing weight, but also due to the large range over which the area varied in the DoE. Kink location, however, is dominant due to the large impact this variable has on the distribution of wing weight between the inboard and outboard components. With the screening test results now justified, the generation of the response surfaces can proceed.

RESPONSE SURFACE EQUATIONS

The eight variables that are the most statistically significant to each of the responses are now used in a second DoE to generate response surface equations for strake weight, inboard wing box weight, outboard wing box weight, and fuel volume available. This DoE is a 3-level central composite uniform, face centered Design of Experiments with 145 cases. Unlike the 2-level DoE used in the screening test, the central composite design captures second order effects as well as interactions, allowing nonlinearities in the response to be modeled by the response surface equations. Again, this DoE is executed with NORMAN and the automation and optimization scheme discussed previously. An ANOVA procedure is then used to generate the coefficients for the response surface equations of the form shown in EQ (1). These expressions are also used to generate prediction profiles which depict the trend of the response to changing design variables. These prediction

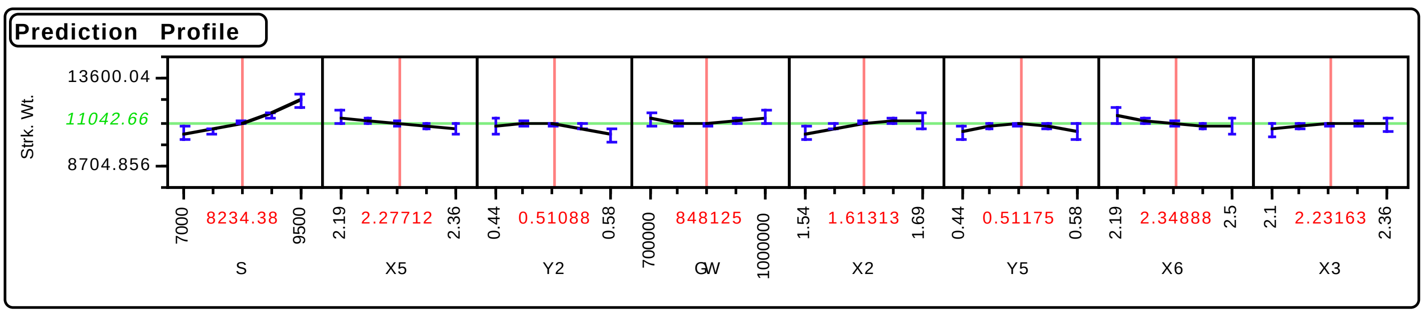

Figure 10: Prediction Profile for Strake Weight

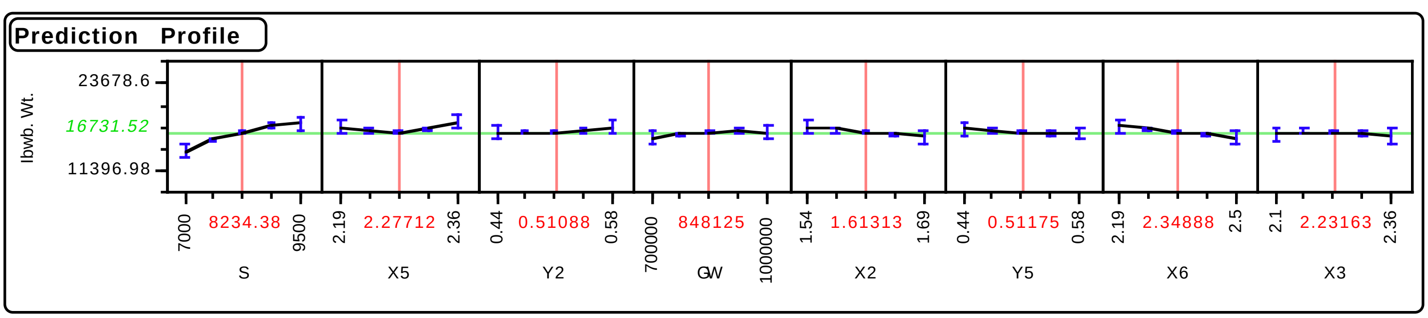

Figure 11: Prediction Profile for Inboard Wing Box Weight

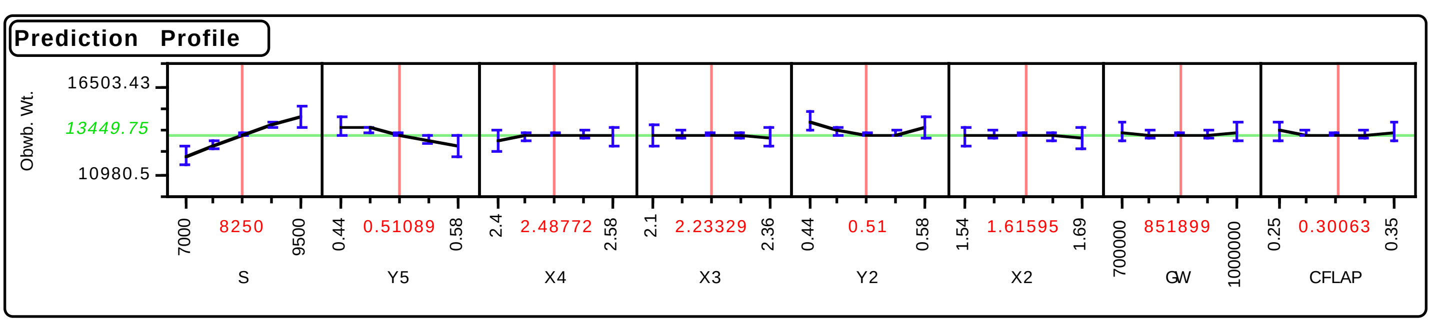

Figure 12: Prediction Profile for Inboard Wing Box Weight

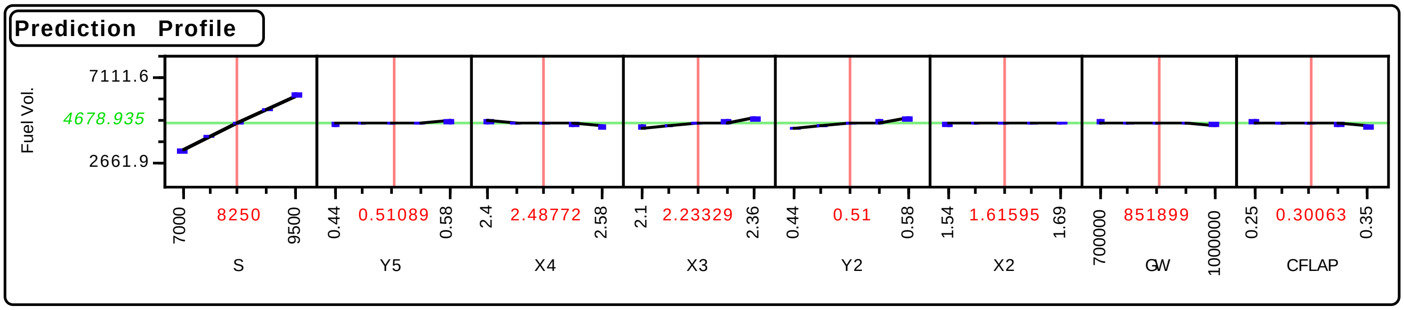

Figure 13: Prediction Profile for Fuel Volume Available

Table III: Material and Structural Concepts for the HSCT Wing with Corresponding Weight Adjustment Factors

	Chordwise Convex beaded all Ti	Spanwise hat-stiffened all Ti	Ti H/C mechanically fastened	Ti Honey-comb welded	Chordwise convex all Ti B/PI spar caps	SPF/DB Ti	PMC Honeycomb	PMC Z-stiffened
Strake	0.871	1	0.867	0.948	0.811	1.072	0.928	1.5
Inboard Wb	1.15	1	0.78	0.798	0.689	1.792	2	2.458
Outboard Wb	1.053	1	0.723	0.744	0.786	0.51	0.569	1

profiles is shown in Figures 10 - 13 for strake weight, inboard wing box weight, outboard wing box weight, and fuel volume available respectively.

As expected, all responses appears to increase with increasing wing area. In addition, the three wing component weights also increases with increasing aircraft gross weight, which directly affects the magnitude of the air load that the structure is sized to for a 2.5 g pull-up maneuver. However, the strake weight appears to decrease with increasing spanwise location of the leading edge kink, which seems contradictory to intuition. These profiles though, show the trend of the response to changes in each variable assuming all other variables are held constant. Therefore, if wing area is held constant and the spanwise location of the leading edge kink is increased, the redistribution of area actually causes the area of the strake region to decrease. As a result, the strake weight will actually decrease with increasing spanwise location of the leading edge kink. The non-dimensionalized x-location of the trailing edge kink, X5, also has a large impact on the strake and inboard wing box weights. As this kink location is moved in the positive x-direction, the strake weight decreases due to the redistribution of wing area to the wing box region. Hence, an increase in X5 will also cause the inboard wing box weight to increase, with all other variables held constant. As these figures illustrate, the prediction profiles generated by the RSM approach provide a powerful visual tool for understanding the relationships between the wing component weights and the design variables.

Response Surface Adjustments

With response surface equations now generated for the weight of the various wing components as well as for fuel volume available, adjustments are now made for non-modeled weight, non-structural weight, and different material and structural concepts. Non-modeled weight is the weight of fasteners or any other material required to complete the structure that is not modeled by ELAPS. Systems in the wing such as fuel tanks and flap actuators are included in non-structural weight. The final factor that affects the wing weight is the use of material and structural concepts such as Super Plastic Formed / Diffusion Bonded (SPF/DB) Titanium or composite materials. Therefore, in order to produce response surfaces which can accurately model the weight of an HSCT wing, the RSE's produced by the scheme discussed previously must be modified for these factors.

First, the non-modeled weight is accounted for by examining a 1975 Lockheed report on supersonic transport wing weights. The configuration modeled in this report is generated in ELAPS and analyzed to determine a wing weight. The ratio of this weight to the weight listed in the report is then used to adjust all response surface results. Therefore, this adjustment is a set percentage of the weight of the wing and is applied uniformly across all configurations, thus preserving the trends of the wing component weights to the design variables.

The weight of the systems within a typical supersonic transport wing is also obtained from this Lockheed report. The ratio of this system weight to the overall weight of each wing component is then determined. Next, similar to the procedure used for the non-modeled weight, this ratio is applied to all response surface results. Again, this ratio is constant across all configurations to preserve the proper trends. The systems included in these calculations are listed in Table II.

Table II: Wing Systems Used in the Determination of Non-Structural Weight (Weights in lb.)

HLFC	8600
Circulation Control	830
De-Icing	415
Engine Support Structure	3580
Main Ldg. Gear Doors	2904
Wheel Well and Attachments	3750
Wing/Body Fairing	1600
Ailerons	1250
Spoilers	1360
Fuel Bulkheads	3800
Total	28089

Although the HSCT wing is modeled as all titanium for the ELAPS analysis, other material and structural concepts for this wing are also being considered. However, concepts such as bismalemide thermoplastics and SPF/DB titanium cannot be directly modeled in ELAPS due to the equivalent plate assumption and modeling within the program. Therefore, adjustment factors are introduced to account for these different concepts. The information for these adjust factors is obtained from the 1974 Lockheed report mentioned previously as well as from the thesis on HSCT wing manufacturing and material concepts[19]. These concepts and adjustment factors are listed in Table III.

Now that the errors from non-modeled and non-structural weight have been minimized, an examination of the error introduced by the response surface approximation to the ELAPS code is completed. Although the R^2 values generated by the ANOVA analysis fall between 0.92 and 0.99, this is only a measure of how well the data used to generate the RSE's fits the resulting polynomials. Therefore, this examination is accomplished by selecting design variable combinations which are not part of the DoE used to generate the response surfaces and comparing the weights obtained from the response surfaces with these variable combination to the weights obtained from direct ELAPS optimization analysis. The results of this examination revealed that the largest error encountered in the wing component weight RSE's is on the order of 3% and is usually closer to 1%. As a result of the adjustments discussed previously and the low error of the response surface approximations, these polynomial equations representing an ELAPS analysis of the HSCT wing to determine wing component weights can now be integrated into FLOPS for the system level analysis.

SYSTEM LEVEL ANALYSIS WITH FLOPS AND FPI

Once the response surface equations for the wing component weights and fuel volume available are generated and adjusted, they are integrated into the FLOPS synthesis and sizing tool. This task is accomplished by replacing the historically based equations for wing component weights and fuel volume with the polynomial response surfaces through direct coding. As mentioned previously, response surface equations describing the aircraft aerodynamics and acoustics are also incorporated into FLOPS to accurately model these characteristics of the HSCT. Propulsion characteristics are modeled through the direct linkage of ENGEN, an engine cycle analysis program, with FLOPS. This direct linkage is used in place of the response surface approach due to the large amount of information which must be passed between ENGEN and FLOPS during sizing of both the aircraft and engine cycle. With the inclusion of the physics-based response surface equations and the direct linkage with ENGEN, FLOPS is now capable of sizing HSCT configurations.

In order for FLOPS to size configurations, however, the mission must first be specified. The HSCT configurations examined in this study are assumed to carry 300 passengers with a range of 5000 NM. This mission is divided into take-off, climb an optimum altitude and cruise supersonically at Mach 2.4, descend to 35,000 ft and cruise at Mach 0.9, and finally descend and land with fuel reserves remaining for a 200 NM detour to an alternate airport at a Mach 0.9 cruise. The subsonic cruise segment is included to account for the fact that the HSCT is not allowed to fly supersonically over land, and therefore must fly a portion of the mission subsonically. This segment is assumed to be 15% of the aircraft range.

Probabilistic Assessment through FPI

In order to assess the take-off gross weight of the HSCT throughout the design space, the FPI technique is used to introduce probabilistic techniques into the design process. This is accomplished by first specifying to FPI the design variables which are to be represented probabilistically. These variables and their corresponding ranges are listed in Table IV.

Table IV: System Level Variables and Ranges

Design Variable	Min. Value	Max. Value
X2	1.54	1.69
Y2	0.44	0.58
X3	2.10	2.36
X4	2.40	2.58
X5	2.19	2.36
Y5	0.44	0.58
X6	2.19	2.50
t/c	0.03	0.05
S_{ref} (ft^2)	7000	9500
$C_{L,design}$	0.08	0.12
Leading Edge Tip x-location of Hor. Tail	0.95	1.73
(Hor. Tail Ref. Area)/S_{ref}	0.045	0.090
(Vert. Tail Ref. Area)/S_{ref}	0.045	0.070
Nacelle Scaling	0.9	1.1
X-location of Wing on Fuselage	0.22	0.28
(flap chord length)/c	0.25	0.35
Overall Engine Pressure Ratio	18.0	22.0
Fan Pressure Ratio	3.5	4.5
Turbine Inlet Temperature (°R)	3000.0	3300.0
T/W	0.26	0.32

It is worth noting that the wing x and y locations on the wing are non-dimensionalized by wing semispan, while the horizontal tail x location is non-dimensionalized by the horizontal tail semispan and the wing x location in non-dimensionalized by the fuselage length. The design variables are assumed to vary uniformly over their ranges, such that any value with the range has an equal probability. In this manner, the resulting cumulative frequency distribution will reveal the probability of the design satisfying the specified constraint. For TOGW, the constraint value is 1,000,000 lb., as discussed previously. Therefore, the technical feasibility, or the ability of the aircraft at a given technology level to satisfy the design constraints, is quickly assessed in a probabilistic fashion.

The resulting cumulative frequency distribution for take-off gross weight is shown in Figure 14.

As this distribution illustrates, the HSCT has only a 20% probability of achieving the TOGW design constraint for the current variable ranges and technology level. Therefore, the feasibility of the design is very sensitive to the wing configuration. This information in invaluable when in the initial stages of design, where major design changes are still relatively inexpensive. However, the evaluation of the design space at a preliminary design level of fidelity usually requires a great deal of time and resources. Through the use of ELAPS, RSM, and FPI, though, this process has been completed in a much shorter amount of computer run time and model generation time.

CONCLUSION

The focus of this study is to illustrate the importance and feasibility of including physics-based structural information through ELAPS in the conceptual design phase. In addition, this study illustrates the use of the FPI technique to assess the design space in a probabilistic fashion.

This study concludes that the current design variable ranges translate to a design space with a TOGW from 925,000 lb. to almost 1,200,000 lb. In addition, the design only has a 20% probability of achieving the 1,000,000 lb. feasibility constraint. These high TOGW not only translate to higher acquisition costs, but much greater direct operating costs as well. Therefore, efforts should be focused toward reducing this TOGW and therefore increasing the probability of the design being technically feasible.

One possible approach is to identify the technologies which can reduce this weight. The analysis presented in this study is for conventional technologies only. When new technologies are introduced into the design, however, the same design variables will result in different, often improved weights. These new technologies might include advancements in materials, or new high lift devices such as circulation control that would eliminate the need for flaps and flap actuators. In addition, improvement in cruise drag from technologies such as hybrid laminar flow control can reduce the amount of fuel required to complete the design mission, and in turn reduce the structural weight. Applications of technologies such as these can lead to improvements in TOGW such as those shown in Figure 15.

As these results illustrate, the application of new technologies can not only bring the probability of achieving feasibility to 100%, but also moves the distribution closer to the economically viable target value of 750,000 lb. Further research should be directed toward the evaluation of the impact of these technologies on technical feasibility through the use of the methods developed in this study. Therefore, this work should be regarded as a first step in order to illustrate an approach for physics-based modeling and probabilistic feasibility assessment. In order to extend this exploratory

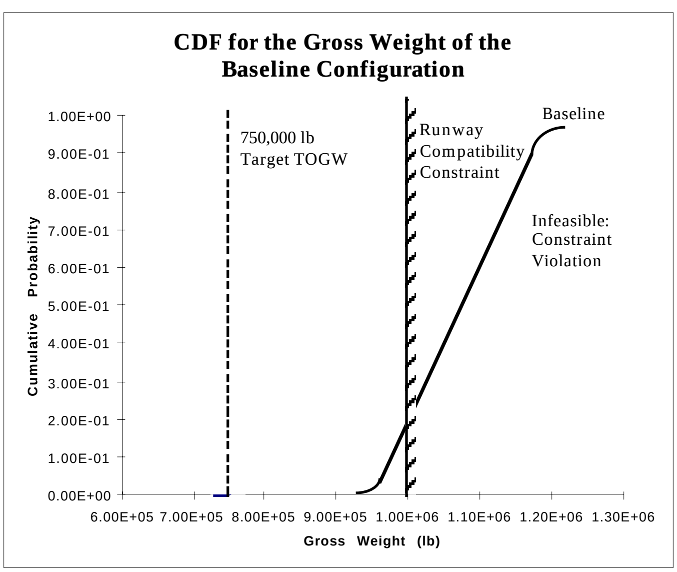

Figure 14: Cumulative Frequency Distribution for Take-Off Gross Weight

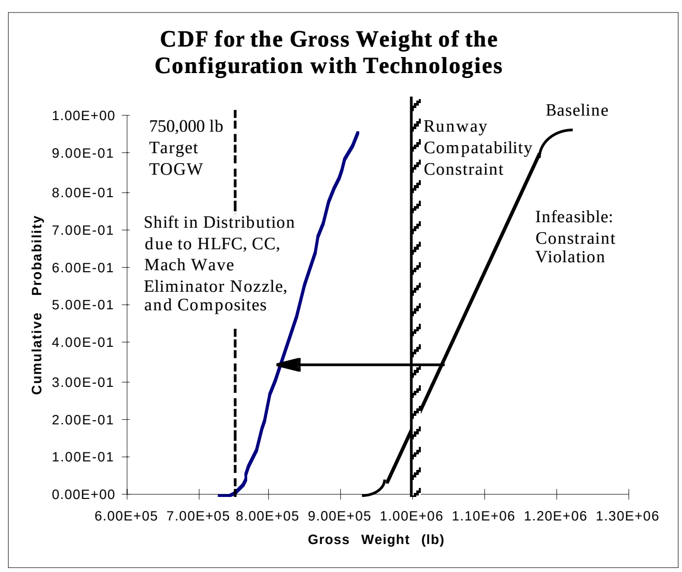

Figure 15: Improvements in TOGW due to the Application of New Technologies

study, future work will center on assessing the economic viability of an HSCT configuration in a probabilistic manner. Additionally, the economic impact of the technologies used to shift the distribution closer to target must be assessed in order to properly measure their benefit.

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